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DESIGN AND TESTING OF ROCKET MOTORS WITH COMPOSITE PROPELLANTS

Dr. SREENADH CHEVULA

Department of Aeronautical Engineering, Mallareddy College of Engineering and Technology, Maisammaguda, Hamlet, Secunderabad, Telangana, India

ABSTRACT

In space missions, satellite launchings and missiles for army applications, solid rocket motors plays a major role. As we know that the thrust produced by the rocket engines only gives the forward motion to the rocket body and its payload. The rockets which used Solid fuel, which contain fuel and oxidizer itself called Solid rocket motor. Solid fuel can be prepared by Composite material powders with various combinations. Each combination contains its own parameters and thrust/weight ratios. Depends upon the requirements of the thrust to lift off the mass of the rocket, specific combinations of the chemicals are used in the preparation of the solid propellant grains. To understand the characteristics of various combinations of the propellants, the present work is mainly focused on testing of solid rocket motor thrust generation under various combinations of composite propellants and burning rates of the same. Small scale experimental testing and the results of positive and negative obtained in more than one test will be discussed and solutions will be provided for the obtained negative results.

KEYWORDS: SRM, Testing of Rocket Motors, Composite Propellants & Thrust Production

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NOMENCLATURE

A_b	Burning Surface Area
ρ	Solid propellant density
M	Mass flow
γ	Burning rate
I_t	Total impulse
I_s	specific impulse
b	web thickness
b_f	Web thickness
n	Burning index
a	Imperial Constant
σ_p	Temperature Sensitivity
π_k	Temperature Sensitivity Pressure
A_t	Nozzle throat area

Original Article

General Subscripts

0	Stagnation Condition
E	Exit
T	Throat
C_c	Combustion Chamber

INTRODUCTION

The historical background of the solid rocket motor started by the investigations of Goddard by use of solids for propulsion, but spent most of his life exploring liquid propellants. Most of his findings had to be rediscovered by others. [1] In the development of tactical missiles, which has been used at the time of World War II were mainly developed at Caltech, West Virginia, and Bruceton, Pennsylvania. [2, 3]. Solid rocket motors were mainly used in this tactical missiles, and as a separate line some researchers from the Guggenheim Aeronautical Laboratory at Caltech worked on them [4]. First time in 1942, in the preparation of a composite propellant for a solid rocket motor booster, Asphalt as a binder and fuel and potassium perchlorate as an oxidizer had been utilized. [4]In the timeline of large solid motors many more discoveries were found out. Various compositions of chemicals were introduced to host a small to huge space launchers. For example, in the building of Polaris A1 missile case designed by 15% aluminum and ammonium perchlorate with the composition of polyester- polyurethane. In 1954, as a result of many experiments by Sutton, it has been identified that PBAA was a good binder with less tear strength. Later To overcome this disadvantage, PBAN has been introduced by Thiokol. As a result of good tear strength PBAN had been the largest production tonnages in the industry [5]. To obtain better mechanical properties in solid rocket motors grains (SRM) Thiokol developed CTPB further this polymer, known as HTPB. [5]

For subsequent missiles and rockets, some questions exist about the development of the propellant grains. Such that did polybutadiene–acrylic acid (PBAA); acrylic acid, acrylonitrile, and butadieneterpolymer (PBAN); carboxyl-terminated polybutadiene (CTPB); and hydroxyl-terminated polybutadiene (HTPB) come to be developed as binders? Did tetra methylene tetranitramine (HMX) will replace at least some of the ammonium perchlorate? Without experimental results based on only technical literature survey these questions are not answerable. In this line more research is needed, and in this paper, some efforts are made to address these questions. Finally, in brief, we can conclude that, a Solid rocket motor (or SRMs) are simple devices with very few moving parts. The propellant contains both fuel and oxidizer; therefore these devices can operate in the vacuum of space. Thrust is developed as the high thermal energy of the combustion gases is converted to kinetic energy in the exhaust [6]. The simplicity of SRMs makes them an attractive choice for many rocket propulsion applications. Because there are few structural components, and most of the weight of the SRM is usable propellant. Concerning disadvantage the I_{sp} for SRM are low compare to liquid propellant. Once SRM got ignited the motor will burn to the end if it don't have a special mechanism to terminate the burning. In the middle of the fire.

In the present work mainly focused on solid rocket motor thrust testing of various combinations of composite propellants, burning rate of the various combinations of the propellants, designing of grains in solid and hybrid rocket motors for high effective thrust and hybrid rocket motor thrust testing with a Bipropellant which requires oxygen on burning process i. e. Thrust production. Experimental testing facility has been established in Pack research cell, Coimbatore, India.

EXPERIMENTAL SETUP

To achieve the goal of the proposed work, with the help of literature and previous experimental setups by the worldwide researchers, the list of equipment's and chemicals have been identified and manufactured for the preparation of propellant and testing. For a small scale of testing facility, those equipment's namely, Micro Mixer, Pressure vacuum casting chamber, Constant temperature hot water bath, Vacuum Oven, Vacuum pump, Decicator, Small Weighing machine, Rocket timer machine, Window Bomb, High pressure regulators.

Figure 1: Micro Mixer Blade

Figure 1 (a): Micro MIXER

Figure 2: Vacuum Casting Chamber

Figure 3: Constant Temperature Hot Water

Figure 4: Vacuum Oven

Figure 5: Vacuum Pump Bath

Figure 6: Decicator

Figure 7: Weighing Machine

Figure 8: Rocket Timer Machine

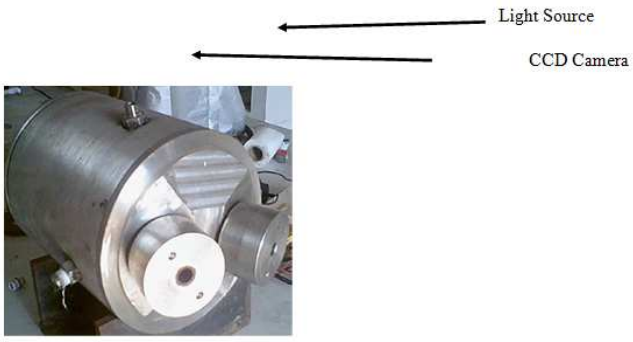

Figure 9: Window Bomb

PROPELLANT PROCESSING AND TESTING

In this section the procedure for the preparation of the propellant has been explained in details. As an initial step in the preparation of the propellant, the combination of the propellant need to be predefined as per the functionality of the chemicals in the final propellant grain and final weight of the propellant. In this work, the combination for the Composite propellant as follows

Table 1: Propellant Mixing Ratio and Weights [16, 17, 18]

S. No	Type of Chemical	Chemicals	*Theoretical %	#Actual %	Theoretical weight	Actual weight
1	Oxidizer	Ammonium Perchlorate	70	69	138	126.8
2	Metal Fuel	Aluminum	30	22	44	42.3
3	Binder	Hydroxyl-terminated polybutadiene	15	8	16	29.3
4	Plasticizer	Dioctyl Adipate	10	2	4	3.8
5	Curing Agent	Isophorone Diisocyanate	1.3	1	2	1.5
				Weight	204	203.7

* Values are the expected % of the chemicals in the final propellant grain
\# Actual % of chemicals availability in the final propellant grain

In the process of propellant preparation initially need to adjust weighing machine in to proper setting given by the manufacturer. Take a beaker of 300 ml and fill the Ammonium perchlorate 126.8g weighted by the weighing machine. After taking the Ammonium Perchlorate tear the weight and add 42.3g of Aluminum powder to it and mix with a stirrer. To this mixer add 29.3 g of HTTB and again mix the propellant mixer with stirrer. After fine mixing add 3.8ml Dicotyl Adepate (D. O. A) to the mixer and stirrer them. Again mix the Isophorone Diisocyanate 1.5 ml and stir them properly.

After getting the rich mixer of the propellant combination next step is to mix entire propellant using a micro mixer (See figure 1, 1a). This mixing process gives a fine mixture of the propellant and total mixing time 45 min to 60 min. After mixing process next step removing the air bubbles or pockets from the propellant to avoid the failure of brining. For this process we need to a content temperature bath with a plunger setting vacuum casting chamber. As we discussed in chapter 4 we have to place the propellant in the chamber under constant hot water supply towards the chamber surface. The temperature in the hot water bath to be maintains 60^{o} to 70^{o}C. The air pockets, removing process takes around 60 to 120 min depends upon the propellant quantity. After removing the air pockets the grain is ready for the curing process which is to be done by a vacuumed oven. The propellant should place in small plastic containers with proper closing. Place the containers inside the oven and close the oven with proper ceiling provided by oven manufacturer. After removing the air from the oven using the vacuum pump we have to switch on the oven and set the temperature 65o -75oC to heat up the inside propellant. This process is known as curing process and the process time 48 to72 hours. I. e. we have to maintain the same temperature continually 48 to 72 hours. After this time our propellant looks like a rubber and ready for testing, and the propellant weight may below compare to the ingredient mixer weight before mixing and curing process. Finally the propellant has been prepared with the expected theoretical weight of propellant 204g, Actual prepared propellant weight = 203.7g, and after mixing and curing process weight of propellant=185g.

Figure 10: Propellant Shape after Curing Process

TESTING OF PROPELLANT BURNINGRATE

In this section the propellant testing process will be discussed for burning rate finder. Burning rate can be found by testing the propellant in window bomb (See figure 9). The main items of the testing process includes Nitrogen gas, High pressure regulator, and 12 volts battery and timer machine (see figure 8)**.** The connection setup of the window bomb discussed in previous section, after wire adjustment done for the testing, the shaped propellant to be prepared for window bomb propellant stand. First of all we measure the total length of the propellant and mark on it for every 5mm distance then insert 38microm Nicrome wire at all marked stations.

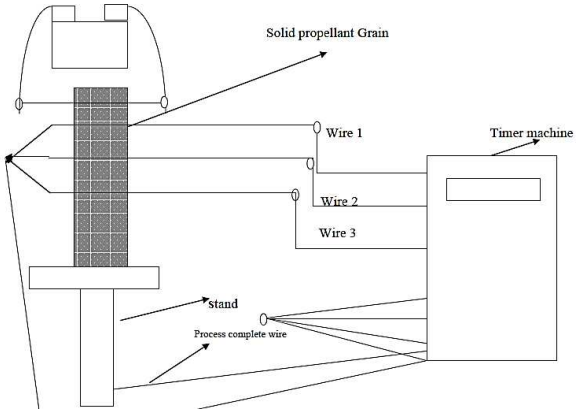

Figure 11: Schematic Diagram of Wire Adjustment with the Sample Propellant

Further Nitrogen gas pressure controlled by with the help of high pressure regulator to send in to the window bomb. The Pressure level of the bomb is noted with the help of pressure gauge which is connected through inlet hose. The first station Nicrome wire is connected with 12V battery electrodes. As we switch on battery Nicrome wire becomes red hot and then it initiates the burning of the propellant. The other wires are connected with the timer machine. As the burning surface precedes it burn the Nicrome wire. Due to burning of wore the connection cut and the timer machine display shows the readings in seconds and mille seconds. We know the distance between 2 wires and time required to burn that distance then we can calculate the burning rate. With the help of the following formula and timer machine readings we can estimate the Burning Rate of the prepared propellant.

Burning Rate= Burning distance / time required to burn (m/s) (1)

For each and every sample we will calculate burning rate for two times with the same pressure. The idea is to compute the result of the same sample but 2 attempts. by this process we will get burning rate near to exact value.

CALCULATION OF THE THRUST

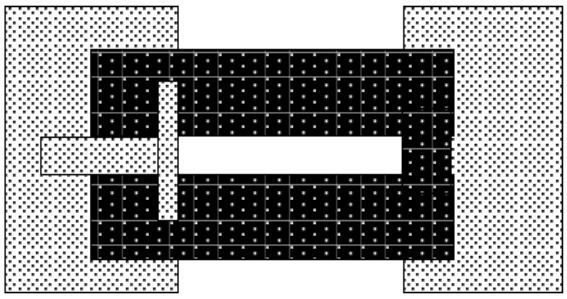

Figure 12: Small Rocket Motor with Tubular Grain [9]

Figure 13: Propellant with Thrust Chamber of the Rocket Motor

Obtained Specifications of the prepared propellant grain and the rocket motor are length of the motor is 60 mm, Gain inner diameter is 5 mm, Gain outer diameter is 26 mm, P_0 = 70 bar, T_0 = 300 K, and (A_e / A_t = 1, Specific Impulse of the propellant is 1041 Ns/Kg at 70 bar, Burning Rate is 6.7 m/s (found from the experimental setup) Molecular Weight is 27.8, Specific Heat ration is 1.138 and Gas Constant is 299.06j/Kg k.

Now the mass flow rate of the gas at starting stage of the rocket = ρ x b xA_b [7]

= 1945 x 6.7 x10-^{3} x π x 5 x 60 x 10^{-6}

= 0.01228Kg/s

$$\frac{M_{max}}{A_t} = \frac{P_0}{\sqrt{T_0}} = \sqrt{\frac{\gamma}{R}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{2(\gamma+1)}}} \quad (2)$$

$$\frac{0.01228}{A_t} = \frac{70\ x10^5}{\sqrt{3000}} = \sqrt{\frac{1.138}{299}\left(\frac{2}{1.138+1}\right)^{\frac{1.138+1}{2(1.138+1)}}}$$

Coefficient of discharge A_t is 4.353 mm^2, D_t is 2.355 mm^2

Mass flow rate of the gas near to end of burning (25mm diameter)

ρ x b xA_b [7]

= 1945 x 6.7 x10-^{3} x π x 25 x 60 x 10^{-6}

= 0.0614 Kg/s

Thrust at rocket Motor Starting stage F =M x I_sx g

= (0.01228/0.60) x 1041 = 21.30 N

Thrust Near to End burn = (0.06141/0.60) x 1041 = 106.54 N

Heat Release at Strat = m_p x C_p x Δt [8]= 0.02047 x3.54 x(3000-288) = 196.5 Kj/s

Heat Release at Near to end of burning = 0.10235 x3.54 x(3000-288) = 982.61 Kj/s

RESULTS AND DISCUSSIONS

The choice of the reference condition for indicating the burn rate is at high pressure 40-70 bar because most propulsion systems operates at high pressure and therefore the behavior at high pressure is more relevant. We are taking the ammonium perchlorate, 68% in the sample, and a sample which contains 0% Aluminum we can get maximum burning rate compared to other combinations. To get exact values of the burning rate is also depend upon the window bomb setup. Before switch on the battery setup should be fully checked, any leakage of nitrogen gas from the window bomb will give variation in result compare to the exact value. The increase in percentage of Aluminum powder will also reduce the Specific impulse; if the specific impulse of the propellant system is high the propellant to be carried will be smaller. This reduces the hardware weight as well. Further if the propellant density is higher the volume required to be store propellant is smaller.

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